U.S. IN-SPACE ELECTRIC PROPULSION EXPERIMENTS

John F. Stocky
Manager, NSTAR
Jet Propulsion Laboratory/
California Institute of Technology
Pasadena, CA, USA

Robert Vondra
University of Dayton/
Phillips Laboratory
Albuquerque, NM, USA

Alan M. Sotton
Phillips Laboratory (AFMC)
Edwards AFB, CA, USA

ABSTRACT
Arcjet and ion propulsion offer potentially significant reductions in the mass of propulsion systems required for Earth orbiting satellites and planetary spacecraft. For this reason, they have been the subject of validation and demonstration programs. After examining the benefits of electric propulsion, this paper discusses the technology base for the Electric propulsion Space Experiment (ESEX) arcjet demonstration experiment and the NASA SEP Technology Application Readiness (NSTAR) ion propulsion validation program. As part of the Advanced Research and Global Observation Spacecraft (ARGOS), ESEX will perform ten 15-min firings of a 30-kW ammonia arcjet.

The National Aeronautics and Space Administration's (NASA's) validation program, NSTAR, consists of two major elements: a g?round-test element and an in-space experiment. The ground-test element will validate the life, integrability, and pert'rmance of low-power ion propulsion. The ir]-space element will demonstrate the feasibility of integrating and flying an ion propulsion system. The experiment will measure the interactions among the ion propulsion system, the host spacecraft, and the surrounding space plasma; and it will provide a quantitative assessment Of the ability of ground testing to replicate the in-space performance of ion thrusters. By involving, industry in NSTAR, a commercial source for this technology will be ensured. Furthermore, the successful completion of the NSTAR validation program will stimulate commercial and government (both civilian and military) uses of this technology.

1 INTRODUCTION

in an effort to increase the payload fraction of satellites and planetary probes, reduce the cost (i.e., size) of launch vehicles, extend the life of satellites, and reduce the duration of planetary missions, two programs have been initiated to demonstrate and validate electric propulsion. One program, sponsored by the United States Air Force Materiel Command, will demonstrate the technology associated with high-power arcjets. The other, sponsored by the United States National Aeronautics and Space Administration (NASA) will validate the technology associated with low-power (<5-kW) ion propulsion technology.

After a brief, quantitative description of the benefits derived from electric propulsion technology, this paper describes the Electric propulsion Space Experiment (ESEX) and NASA SEP Technology Application Readiness (NSTAR) validation programs.

2 IMPORTANCE OF ELECTRIC PROPULSION FOR MILITARY MISSIONS

2.1 Military Needs

Advanced propulsion technology for military needs dots not differ in kind from that for commercial and NASA spacecraft but it does differ in degree. For any satellite, it is desirable to reduce the mass of the cm-board propulsion system to increase the functionality of the satellite. In addition, increased propellant efficiency (i.e., higher specific impulse) can be used to carry additional propellant, thereby extending satellite life or increasing the scope of work done by the propulsion system, e.g., repositioning.

If increased satellite capability were desired, using ion propulsion instead of chemical propulsion would increase the mass that could then be used for additional payload. For example, a commercial communications satellite could use this additional mass to increase the number of transponders carried by the satellite. Or a military satellite could use the increased mass to enhance communications capabilities by flying larger aperture antennas.

To reduce the cost of a space mission, it is desirable to use the smallest launch vehicle possible. Because ion propulsion can reduce the mass of the required on-board propulsion system dramatically, it may be possible in some cases to combine ion propulsion for on-board use with an ion propulsion module for low-Earth mbit (LEO)-to-geosynchronous Earth orbit (GEO) transfer and reduce the launch vehicle size required for a given spacecraft capability from a Titan to an Atlas.

2.2 Benefits

2.2.1 *Station Keeping*

Station keeping of a GEO satellite requires 49 m/sec ΔV annually for north-south station keeping and 2 m/sec ΔV annually for east west station keeping. The larger the satellite and the longer it remains in orbit, the more efficient the on-hoard propulsion system must be in its use of propellant. The measure of this efficiency is specific impulse (I_{sp}). Compared to on-board chemical systems, electric propulsion increases I_{sp} by factors of **2 to 4** when using arcjets and of 10 or more when ion propulsion is used.

To obtain these benefits with electric propulsion, the propulsion system dry mass must be increased. This increase in dry mass requires a propellant conditioning unit not required by a conventional chemical propulsion system. This increased dry mass means that the propulsion requirement must exceed a certain minimum before electric propulsion demonstrates a performance advantage relative to chemical propulsion (Figure 1).

2.2.2 *Repositioning*

Repositioning refers to changing a GEO satellite's longitude so that the area on the Earth's surface can be viewed by satellite sensors and antennas. Electric propulsion can accomplish repositioning maneuvers more efficiently than chemical systems and in less time (Figure 2). Because electric propulsion uses less propellant during a satellite repositioning performed at a specified rate, electric propulsion can extend a satellite's life and reduce the wet mass required for the propulsion system. "l'his point is rnadc in Figure 3, in which the wet mass of the on-board propulsion system needed for station keeping anti for a 90-deg, 30-clay

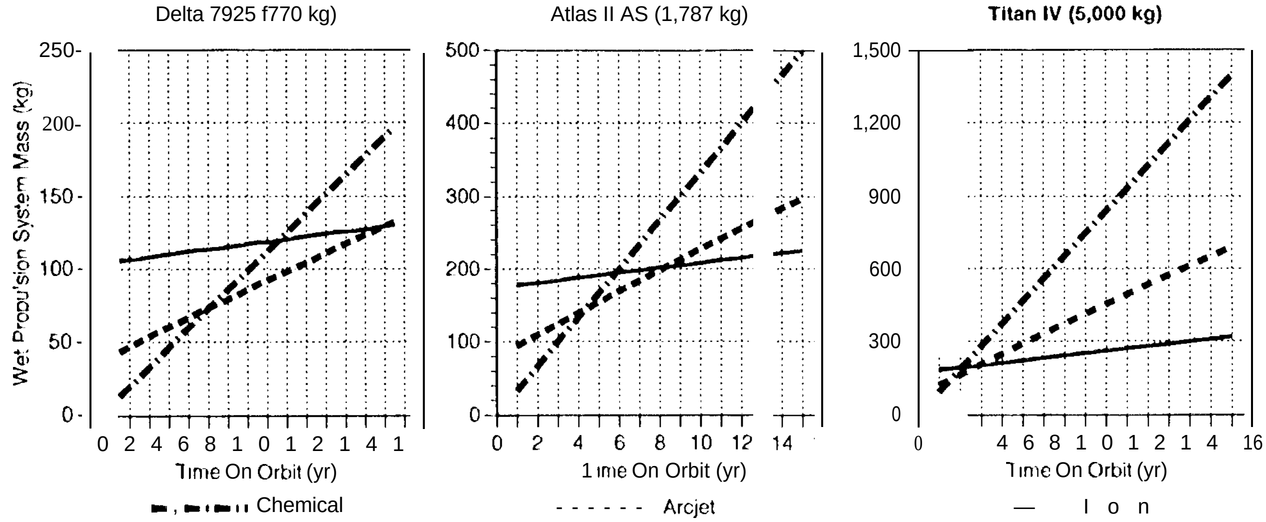


Figure 1. Comparison of station-keeping performance of chemical, arcjet, and ion propulsion.

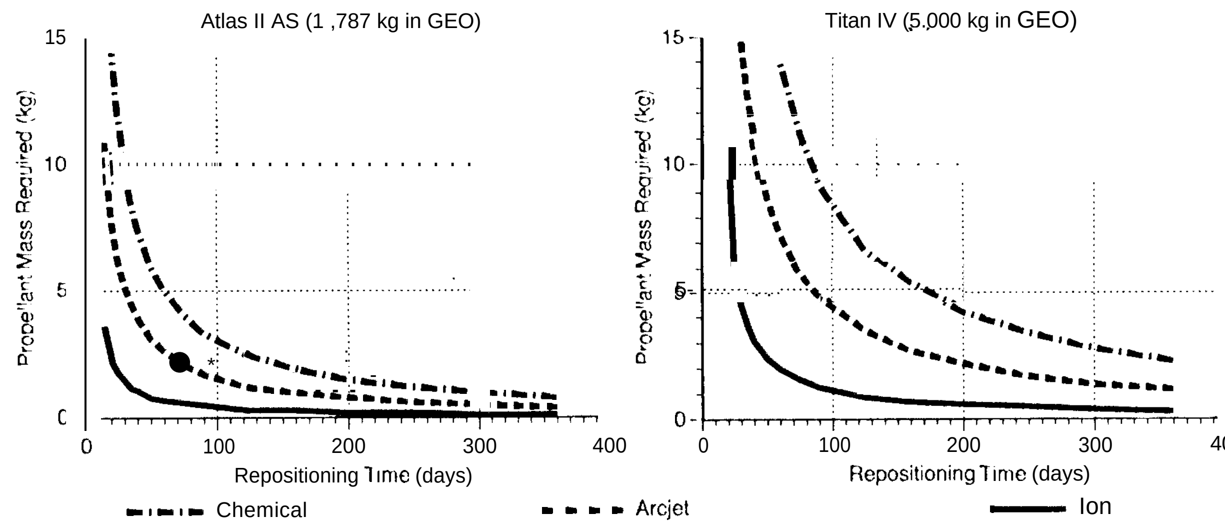


Figure 2. Propellant mass versus repositioning rate for a single 90-deg reposition.

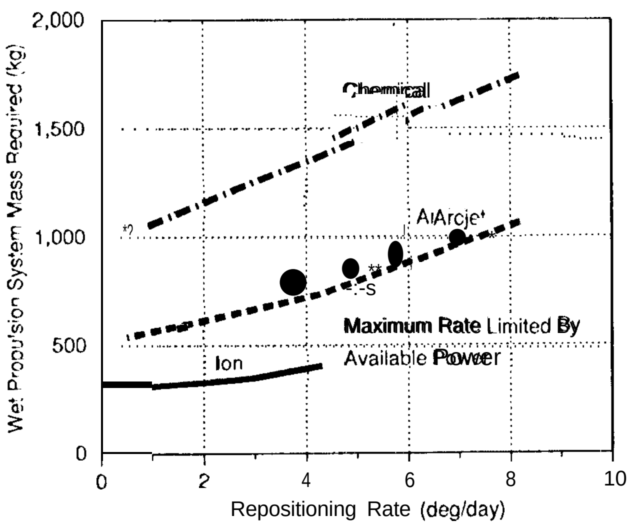


Figure 3. Repositioning performance comparison (for 10 years).

reposition of a GEO satellite with an initial mass in GEO of 5,000 kg and a life of 10 years is shown for chemical, arcjet, and ion propulsion systems. Figure 4 shows a comparison of propulsion systems calculated for Delta-, Atlas-, and Titan-class payloads as a function of on-orbit lifetime for as long as 15 years, assuming two 90-deg/30-day repositions per year. Naturally, the larger the satellite and the longer it remains in orbit, the larger the total impulse required and the more advantageous the higher specific impulse that electric propulsion systems can provide.

We assume that it is more important to increase the number of spacecraft maneuvers than it is to increase maneuver speed. This increase in maneuvers increases satellite life and operational flexibility. Currently, chemical systems nominally carry sufficient fuel for three 180-deg maneuvers. Five-deg/day maneuver rates are nominal, and 15-deg/day rates are reserved for crisis maneuvers. The mass of the chemical propulsion system (fuel and dry mass) is calculated for a range of spacecraft maneuvers at different rates. System masses for ion and arcjet systems (includ-

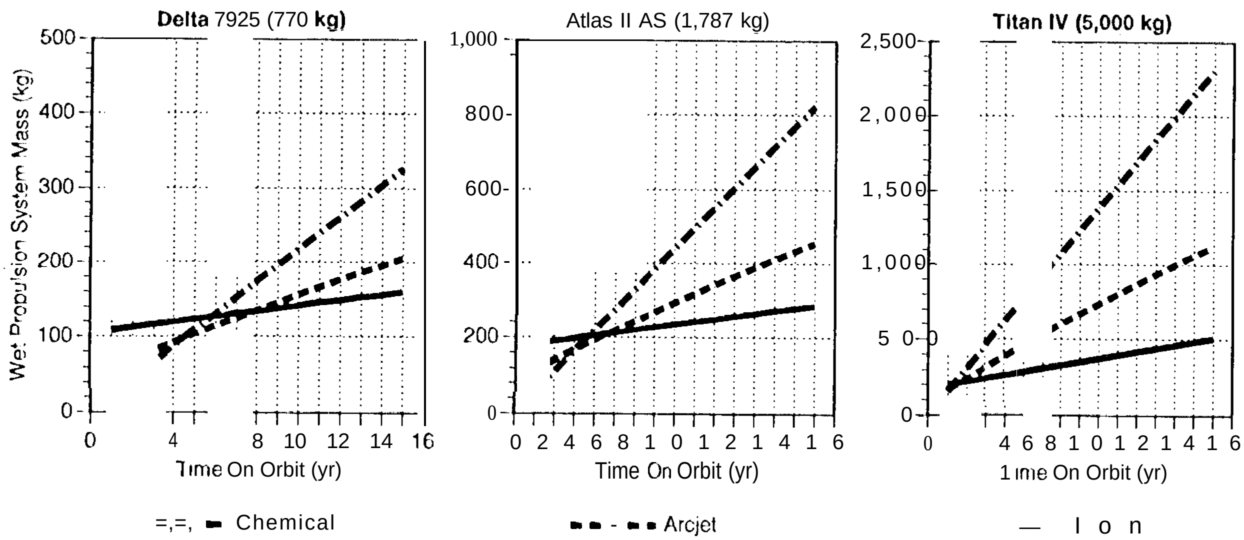


Figure 4. Station-keeping and performance comparison for chemical, arcjet, and ion propulsion.

Table 1a, Assumptions for calculations of electric propulsion performance.

System	Mass, Basis	Specific Impulse, lb_f-sec/lb_m	Tank Fraction, %	Efficiency, %
Chemical Propulsion, Dry Mass	o kg	220	10	N/A
Arcjet Propulsion, Specific Mass	5 kg/kW	7(KI	10	35
Ion Propulsion, Specific Mass	10 kg/kW	3,500	10	70

Table 1 b. Spacecraft characteristics.

Spacecraft Mass, kg	Solar Array Specific Mass, kg/kW	Spacecraft Power, kW
909	20	1, 2, 3
2,270	20	1,5, 10,30
4,s50	20	5,](), 20, 30

NOTE: Maneuver = 180 deg at 5 deg/day; the number of maneuvers performed by a chemical system = 3; the electric propulsion system wet mass was set equal to that of the chemical system.

ing solar arrays) arc set to equal the chemical system mass. The fuel mass component is calculated, and the number of maneuvers is found and compared to that of the chemical system.

For a given power, electric propulsion can perform a range of maneuvers dependent cm fuel consumption. Maximum maneuver rate occurs at maximum fuel consumption, i.e., when the thrusters are operated continuously. Therefore, for a fixed power, the rate can be increased by increasing thruster on-time at the expense of the fuel mass pcr move. Also, the maneuver rate can be increased by increasing power to the thrusters, which increases the solar-array mass and decreases the total fuel that can be carried. Thus, for a given maneuver rate, there is a power level that minimizes fuel mass per move and maximizes the number of maneuvers. As an example, consider two cases in which 1) the solar array is part of the electric propulsion system and 2) the solar array is not part of the electric propulsion system. For this analysis, the assumptions shown in I'able 1 (a, b) were made.

The results of these analyses are shown in I'able 2, in which the power requirements for the smallest spacecraft considered are a modest 1 to 2 kW. An ion engine that carries its own power can execute two to three times the number of manuevers than a chemical system can. If the ion propulsion system is not charged with the power mass, the number of maneuvers increases by a factor of 10 over chemical. However, 1 to 2 kW is not enough power for an ion propulsion system to execute a crisis respositioning of 15 deg/day.

For the 2,270-kg spacecraft, the power requirements are 1-S kW. The ion engine increases the number of maneuvers by a factor of up to 10 over chemical propulsion. If the spacecraft has 10 kW of power, ion propulsion provides 1 S deg/day maneuvers. The arcjet requires 5 kW.

In the case of the 4,550-kg spacecraft, the power requirements arc 5-10 kW. When the ion engine carries its own power, it provides

Table 2. Power requirements for arcjet and ion propulsion systems with spacecraft masses of 909 kg, 2,270 kg, and 4,550 kg.

Spacecraft Mass, kg	Reposition Rate, deg/day	Arcjet				Ion Propulsion				Prop. Wet Mass, kg
		With Array		No Array		With Array		No Array		
		Number of Moves	Power, kW	Number of Moves	Power, kW	Number of Moves	Power, kW	Number of Moves	Power, kW	
	5	3	1	7	1	9	1	26	1	39
909	10	5	1	6	1	7	2	24	2	78
	15	4	2	6	2	Power Limited		Power Limited		117
	5	5	1	7	1	22	1	29	1	98
2,?70	10	3	5	6	5	7	5	24	5	196
	15	4	5	6	5	Power Limited		20	10	294
	5	3	5	7	5	8	5	26	5	197
4,550	10	5	5	6	5	8	10	24	10	392
	15	4	10	6	10	Power Limited		20	20	586

twice as many maneuvers as a chemical propulsion system. When the ion engine dots not carry its own power, the number of maneuvers increases by a factor of 9 over chemical propulsion. With 20 kW of power available, the ion propulsion engine provides 1S deg/day maneuvers. The arcjet requires 10 kW.

2.2.3 *Communications*

Alternatively, the reduct ion in wet mass of the propulsion system can be used to increase the functional capability of the satellite. As an example, the size of an antenna that can be carried by a GEO satellite can be estimated [0 determine if a larger antenna can be carried. As a starting point, the non-propulsion mass of a GEO satellite having a chemical, on-board propulsion system capable of performing not th south and cast- west slat ion keeping and two 90-deg/30-day repositions per year for 15 years was calculated. This non-propulsion mass was taken to be a measure of the satellite's functional capability, its "functionality," and was held constant to ensure that the satellite's capability was not compromised. Added to this payload mass was the mass of the ion propulsion system required for the same station-keeping and repositioning functions described above for a chemical propulsion system. The difference between this sum and the mass that could be placed in GEO by the launch system was calculated for each year of the satellite's life, and the diameter of a rigid antenna having a mass equal to this difference was estimated. The results are shown in Figure S for Titan and Atlas launch vehicles.

The scaling equation for the antenna was

$$\text{Mass (Antenna)} = 4.7471) - 4.61\ D^2 + 1.793\ D^3$$
$$D = \text{Antenna Diameter (m)}$$

The results shown in Figure 5 indicate that ion propulsion would allow a GEO satellite to carry an antenna larger than 10-m diameter (Titan launched) or 7-m diameter (Atlas launched) and still retain a long on-orbit lifetime.

?.?.4 *Orbit Transfer*

Significant savings can be realized if electric propulsion can reduce the initial mass of a GEO satellite so that it can be launched with a smaller launch vehicle without changing the functionality of the satellite. Figure 6 shows that the current performance of both ion and arcjet propulsion systems dots not adequately reduce mass. The question then arises, what LEO-to-GEO transfer time would be required if, in addition to employing an on-board electric propulsion system, an electric propulsion system were used from LEO to GEO? To answer this question, we assume the following: a GEO satellite with a 15-year life requirement and an on-board ion propulsion system able to support north south and cast- west staticm-keeping requirements and two 90-deg/30-day repositions performed annually. The satellite also provides a functionality equivalent to that of a IS-year GEO satellite using a chemical propulsion system able to satisfy the same staticrn-keeping and repositioning requirements.

For this scenario, the mass of solar-powered electric propulsion transfer modules was calculated, We assume the system would use either an APSA-type solar array with GaAs solar cells or a concentrator array. For the ion propulsion system, the performance being validated by NSTAR was assumed; for the arcjet system, a currently available system using ammonia was the basis for one set of calculations and an advanced system using liquid hydrogen was the basis for the other set of calculations. For each launch vehicle considered, the mass of the satellite was subtracted from the launch vehicle's lift capability to LEO. The remainder was used for the solar-powered electric propulsion system. The

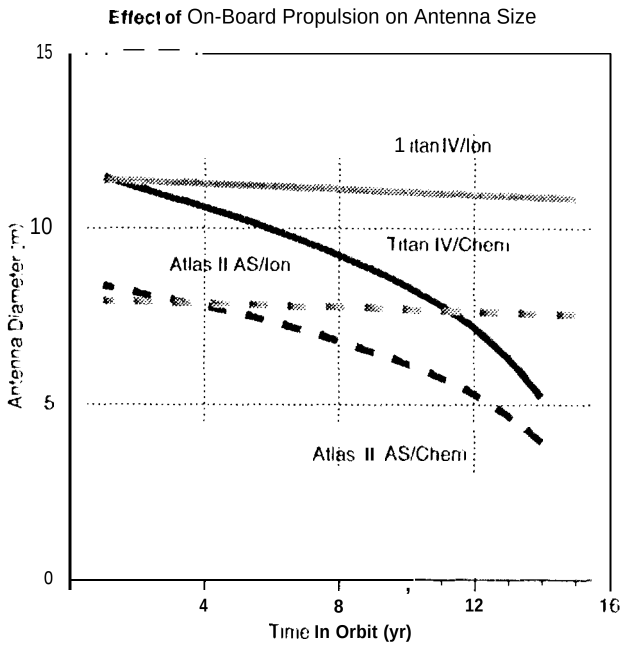


Figure 5. Antenna size is influenced by the choice of on-board propulsion technology.

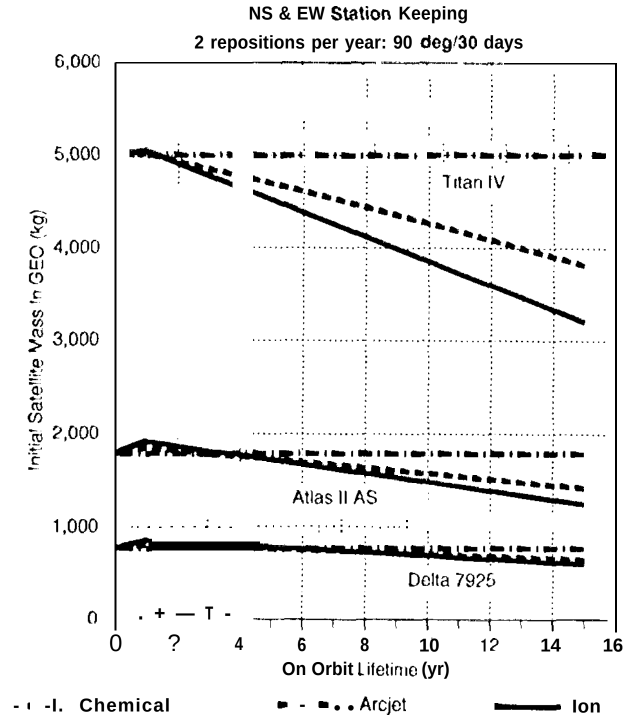


Figure 6. Reductions in mass are derived from advanced on-board propulsion in GEO.

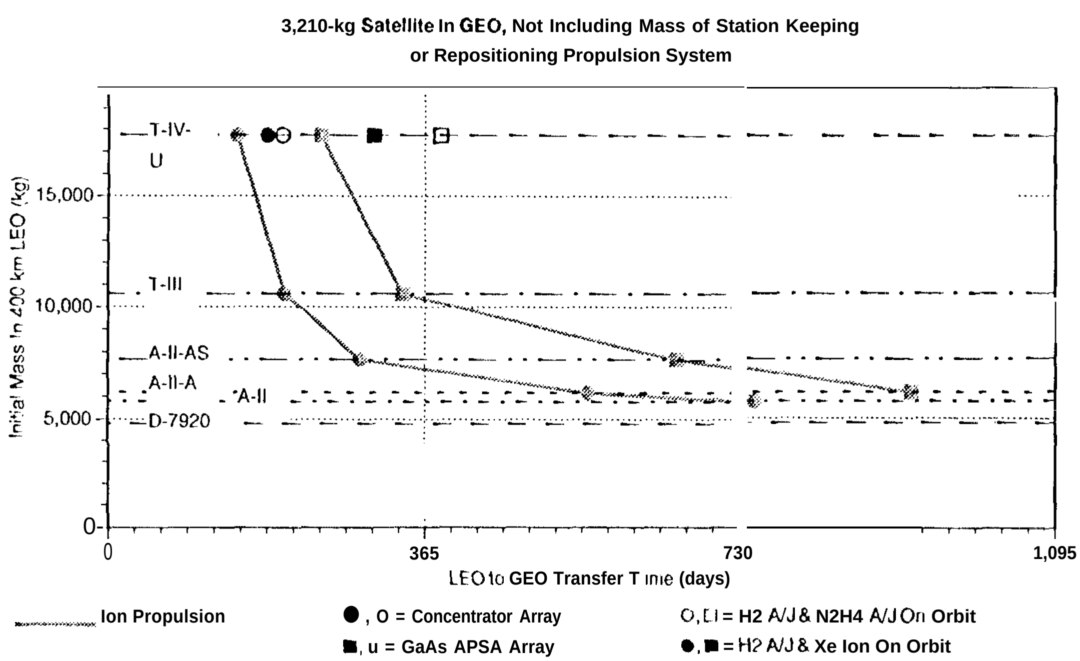


Figure 7. Initial mass in LEO versus LEO-to-GEO transfer time for constant payload mass.

larger launch vehicles would allow a higher powered electric propulsion module and would result in shorter LEO-to-GEO transfer times. The transfer times were then calculated and plotted in Figure 7; this figure shows that today's ion propulsion could he used to place a Titan-IV class payload into GEO using an Atlas or Titan-III launch vehicle with a 6-month to 2-year transfer time. **According, to these calculations, the size of the launch vehicle could not be reduced when** using arcjet propulsion modules.

Whether the savings in launch vehicle cost and launch campaign duration outweigh the penalty of a lengthy LEO-to-GEO transfer time can only be answered in the context of a specific mission and, consequently, cannot be discussed here.

The results presented in Figure 7 assume a constant spacecraft design technology level. If we consider the advances associated with equipment and instruments made by such programs as the

Space Defense initiative (S111), the results shown in Figure 7 may be unduly conservative.

3 ESEX

3.1 Arcjet Technology Background

The Phillips Laboratory has been developing arcjet technology foi a number of space applications - originally for orbit raising, (ESEX and Electric Insertion Transfer Experiment (ELITE)) and more recently for orbit repositioning (modified ELITE). (ESEX is a flight program that will be discussed later.) The ELITE program was canceled, but the arcjet technology and repositioning application that stemmed from it are worthy of study.

3.2 Arcjet Technology Development and Testing

3.2.1 30-kW Arcjet

The Phillips 1 aboratory began development of 30-kW ammonia alc'jet technology in 1984. The Space Defense Initiative contributed to the project by funding technology development centered on endurance and performance testing of promising designs. "l'his work was the foundation for the ESEX arcjet design. After changes were made to the cathode, constrictor, and nozzle, the best designs were endurance-tested at 30 kW. The tests ran less than 500 hours (the goal was 1,500 hours, commensurate with orbit-raising requirements). Performance testing yielded a specific impulse of 754 sec and 29% efficiency. The primary failure mechanism was cathode whisker growth that shorted the electrode gap ending operation. The cathode erosion rate seemed to support 1,5(K) hours of operation (Ref. t). Rocket Research improved on Ibis design in the ESEX program and demonstrated 8 t 5-see specific impulse at 30% efficiency. Lifetime was not addressed.

3.2.2 10-kW Arcjet

The 30-kW arcjet was throttled and performance was measured over a range of powers down to 1() kW. Operat ion was stable and performance was acceptable over this range. As power was reduced, specific impulse decreased and efficiency increased to >600 scc and 37%, respectively, al 10 kW, ELITE's maximum operating level. Two endurance tests were then performed, one at 10-kW continuous operation and the other a cycled on/off operation at 10 kW.

The first test ended after 1,460 hours of continuous operation. The computer shut down the test when it detected a rise in vacuum tank pressure. When examined, it was discovered that the arcjet boron nitride backplate had cracked, causing, propellant to teak. The electrodes, however, were in excellent condition, and there were no signs of whiskers. The pointed port ion of the cathode tip was flattened; otherwise tbc conical section was fully intact. The anode showed no apparent signs of erosion, and the constrictor region seemed unaffected. This demonstration represented 50% more li fetime than required for ELITE (Ref. 2).

Because the ELITE mission required 540 on/off cycles, a 10-kW cycled test was conducted with the arcjet on for crnc hour, off for one-half hour and repeated indefinitely. The test ended after 707 cycles because of vacuum chamber facilit y problems, which were believed to be caused by the arcjet. Rather than destroy evidence by turning the engine cm and risking damage, the test was stopped. When the engine was disassembled, the arcjet was found to be in good credit ion. 'l'he thruster displayed 31% more cycles than required for ELITE. Performance (specific impulse, 62(J 640 sec and thruster efficiency, 33.5 34.5%) was lower than during the continuous test, and the erosion rate (cathode loss, 0.31 g) was higher (Ref. 3).

Next, the arcjet design was modified and its performance characterized over a 3-10-kW range, which is the operational range expected on ELITE as the arrays pass through the Van Allen belt, The cathode gap was shortened (from 0.240 in. to 0.080 in.), and the constrictor diameter decreased (from O. I SO in. to 0.100 in,). The best performance design (specific impulse, 600 700 scc and efficiencies greater than 30% over 3 JO kW) was selected and tested in an integrated system that simulated the solar array-arcjet subsystem being designed for ELITE.

The integrated test-bed consisted of a solar-array simulator and peak power tracker provided by TRW, Inc., a NASA Lewis Research Center's (LeRC's) power processor unit that powered the elect{ ic thr uster, and a Jet Propulsion Laboratory (JPL)-designed ammonia arcjet. The solar-array power source first turned on the arcjet. Oncc the arcjet was ignited, the power to the arcjet was raised to the desired level and operated at this level for a predetermined time. If the power deviated from its maximum value, it was quickly corrected by TRW's electronics. The output of the solar-array power source was then changed, and the process was repeated until the arcjet system was tested over a specified range of interest, which for ELITE was 3 10 kW.

These tests proved proper arcjet ignition and the ability of the system to operate dependably. When the operating power point was intentionally moved off its maximum value, TRW's electronics responded within a second to return it to its maximum value.

3.3 ESEX Program Description

Currently, the Air Force Materiel Command's Phillips Laboratory is developing an arnnmnia-fueled arcjet propulsion system that will be flown as the Electric propulsion Space Experiment. ESEX is being built by a team consisting of researchers from TRW, Inc., Olin Aerospace Corporation (OAC), and CTA (formerly DSI) (Figure 8). ESEX will be the first on-orbit demonstration of a high-power (30-kW) arcjet propulsion subsystem. After 100 hours of battery charging, ESEX will fire the arcjet propulsion subsystem 10 times each for a duration of 15 min (a total of J50 rein).

.7..7.1 Objective

The ESEX experiment has two major objectives: The first is to develop a reliable flight arcjet system and successfully complete a test firing in space, verifying the system's performance. The second objective is to gather data on key spacecraft integration issues, verifying that a bip,h-power arc plasma source can operate without adversely affecting, a spacecraft's nominal operations (Ref. 1).

The major hardware components include a high-power arcjet, Power Conditioning Unit (PCU), and ammonia Propellant Feed Subsystem (PFS). These components were flight-qualified by vibration testing, thermal-vacuum testing, and a 150-min life test. All components were tested as an integrated system in order to gather ground-performance data (Ref. 2). These data will be compared to the flight performance data, which include thrust, specific impulse, and arcjet efficiency. Thrust will be derived by

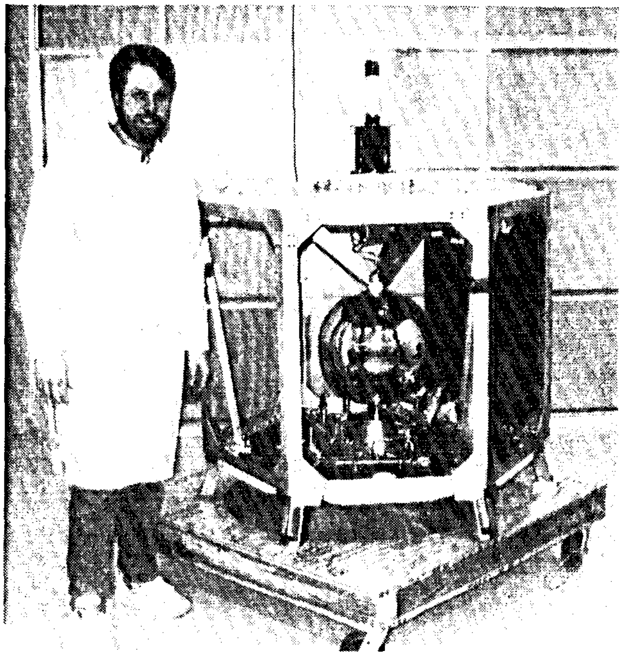


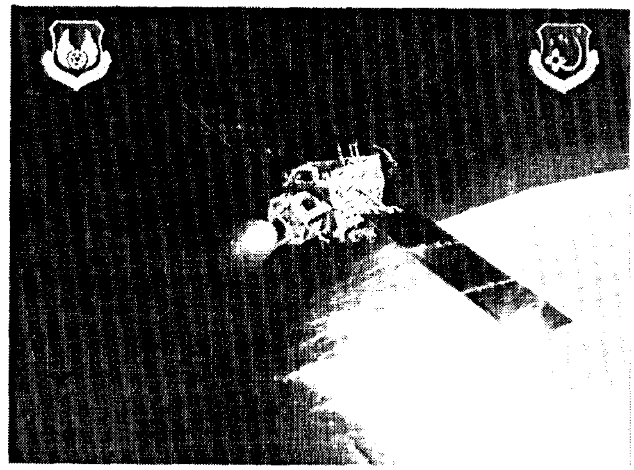
Figure 8. ESEX flight experiment.

combining the spacecraft mass with an accelerometer measurement. Specific impulse will be determined from the propellant mass flow rate and thrust. Efficiency will be derived from the voltage current product (power) and the thrust data. Because electric propulsion devices historically have been encumbered by ground-facility errors, comparable flight data are needed (Ref. 3). The electric propulsion spacecraft interactions that most concern designers are electromagnetic interference (EMI), plume contamination, and thermal radiation. However, it is difficult to measure plume contamination and EMI accurately in ground facilities because the vacuum chamber walls can greatly affect these measurements.

A higll-power arcjet operating at hundreds of amperes of current is a potential source for EMI (Ref. 4). Spacecraft designers can work around EMI, but first they must characterize it, The 1 ESEX antennas will measure EMI in the GHz-frequency range, which corresponds to satellite communicat ion channels.

During life tests of the arcjet, it was discovered that tungsten was lost from the electrodes. Tungsten represents a ser ious contamination issue for solar arrays and optics. However, it is assumed that this mass is ejected away from the spacecraft at a velocity close to the arcjet exhaust velocity. ESEX will measure the deposition of tungsten and other contaminants impinging on the spacecraft to ver ify this assumption.

The arcjet converts approximately 30% of its energy into thrust, '1'here'fore, about70% of the total energy is either conducted to the spacecraft as heat or is test into space (by radiation and frozen flow losses). Although conducted heat loss can be measured on the ground, the portion of the expelled energy that is radiated back to the spacecraft from the arcjet plume cannot easily be measured in ground tests. Radiated heatis affected by plume size and shape, which is determined by the background pressure and vacuum-chamber geometry. ESEX will be ab]e to measure the amount of thermal radiation impinging on the spacecraft during a firing



3..7.? *Host Vehicle*

ESEX is one of eight experiments scheduled to fly on the P9 I -1 spacecraft, the Advanced Research and Global Observation Satellite (ARGOS) (Figure 9) in early 1996. ARGOS is managed by the Space Test and Experiment Programs Office at the Space and Missile Systems Center (SMC). ARGOS is being built by Rockwell International and will be launched by a Delta II into a 460-nautical mile, 98.74- deg inclination orbit (Ref. 6). in addition to the measurements that will be made on board ESEX, ground controllers will be monitoring and recording the ARGOS state of health. in the event that the arcjet adversely affects Al<GOS, the firing will be terminated, However, because of ARG OS' robust design and the fact that arcjet operation is not mission essential, the ESEX experiment offers little risk to the host satellite.

3.3.2 Schedule

In May 1994, ESEX completed component flight qualification and delivery (Figure 10a). Integration was completed in July, and harness fabrication was completed in August. System flight qualification began in September. Delivery to SMC for integration into ARGOS is scheduled for Febr uary 199S. ARGOS is currently scheduled for launch in January 1996 (Figure 10b).

4 THE NSTAR PROGRAM

In 1993, prompted by a request from the USAF/Phillips Laboratory (USAF/PL) to participate in the ELITE program, NASA i nit iated a program to validate low-power irrn propulsion technology. This program, funded jointly by the NASA Office of Space Science and Office of Space Access and Technology, became the NSTAR validation program.

For NASA, two major benefits could be realized once the NSTAR program was completed. First, for small-body rendezvous and planetary flyby missions, ion propulsion would allow NASA to usc Delta-class launch vehicles rather than Atlas- or Titan-class launch vehicles. With ion propulsion, the Delta-class launch vehicles could per form comparable m even enhanced missions (Figure 1 1).

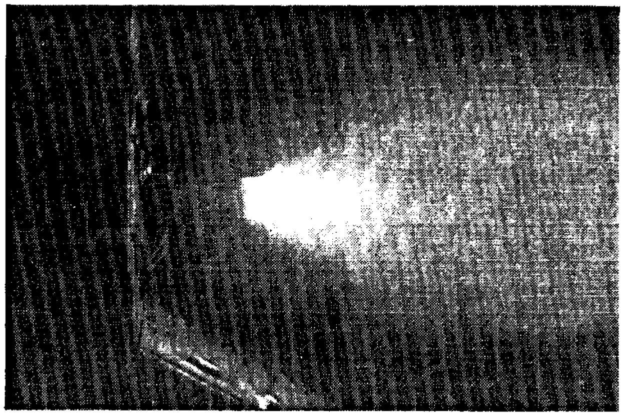


Figure 10a. 30-kW Arcjet (ESEX).

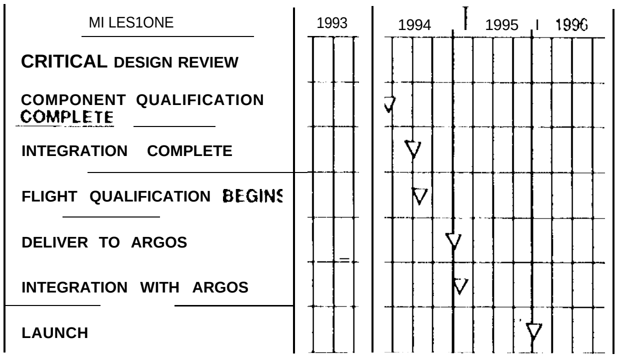


Figure 10b. ESEX schedule.

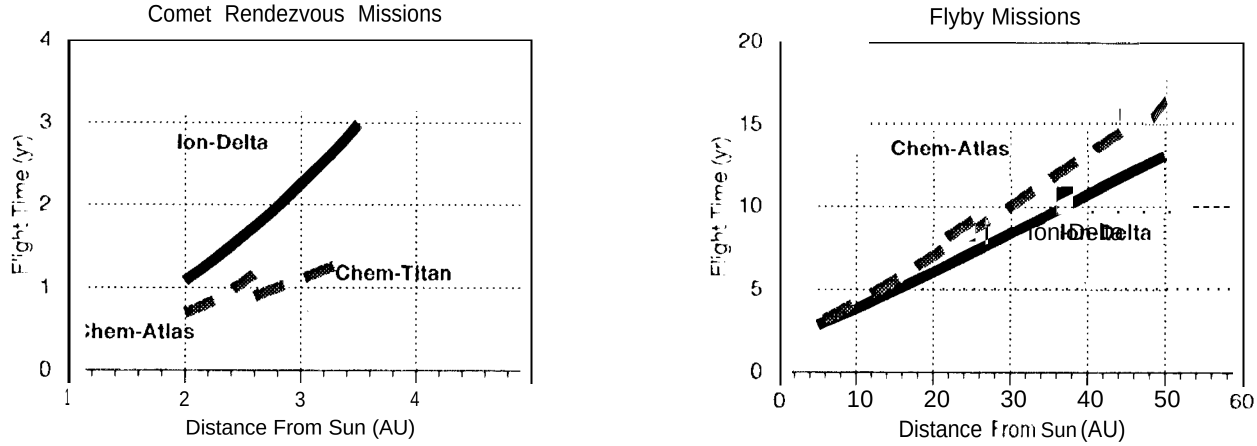


Figure 11. Comet rendezvous and planetary flyby performance comparison.

Second, using a small ion propulsion system with specific impulse ten times that of a chemical system would improve significantly the life or performance of large satellites in geosynchronous Earth orbit. For military satellites, an on-board ion propulsion system would increase the satellite's ability to reposition itself without compromising its on-orbit lifetime; it would still weigh less than the chemical system it replaced, which is sized only for stationkeeping of a large GEO satellite.

Studies show that for each application a single ion propulsion system is required. The system is composed of a 30-cm ion thruster, operating at a power level of 2.5 kW (input to the power processor) with a full-power lifetime of 8,000 hours, and a power processing unit with an efficiency of 92%.

An ion thruster (see Figure 12 for a schematic view) ionizes a propellant (xenon), accelerates the ions through a voltage drop (on the order of 1,000 V), and neutralizes the departing ions with electrons from a neutralizer. Like a chemical propulsion system, an ion propulsion system has a thruster and feed system and requires a power source and power processor to provide the thruster with power at the required DC voltages (Figure 13).

After years of development, the components of an ion propulsion system (ion thrusters, power processors, miniature feed system components, solar arrays, and distributed computer controls) are ready for validation and application on a spacecraft. The development of ion propulsion technology coincides with efforts to reduce the costs of space missions. When deciding, to invest in a space mission, today all costs including the costs of launch vehicles and post-launch mission operations and data analysis (MO&DA) are considered, and ways to reduce these costs are a major consideration. This focus on reducing costs has served to highlight the benefits of ion propulsion - a technology that can shorten mission duration, reduce the costs of MO&DA, and allow spacecraft to be launched with smaller launch vehicles, which would not be possible if chemical propulsion alone were used.

4.1 NASA's Ion Propulsion Verification Program

4.1.1 *Overview*

Ion propulsion offers a way to use smaller launch vehicles and still reduce trip time for a broad class of planetary missions. At the same time, ion propulsion can significantly improve the performance of large commercial and military satellites in GEO. Because of the benefits ion propulsion can offer, NASA initiated the NSTAR program to validate low-power ion propulsion technology.

4.1.2 *Purpose*

The purpose of the NSTAR program is to obtain information that would allow a Project Manager to baseline ion propulsion for a spacecraft.

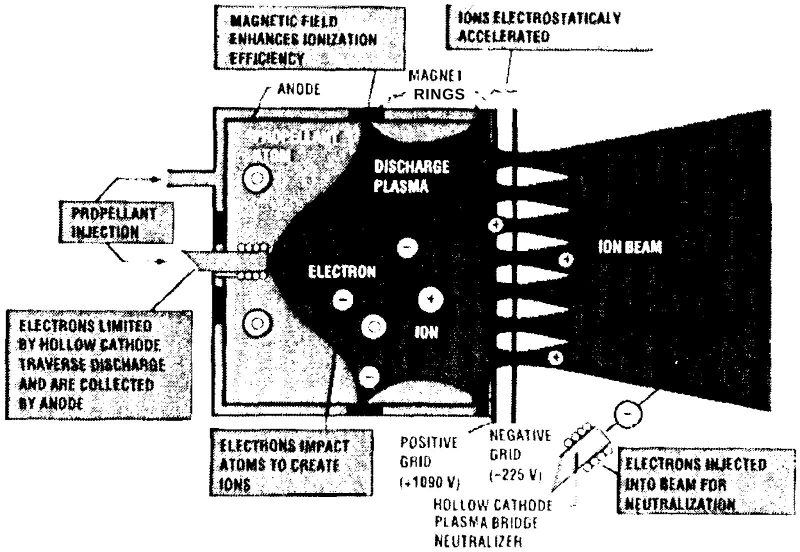


Figure 12. Operation of a gridded ion thruster.

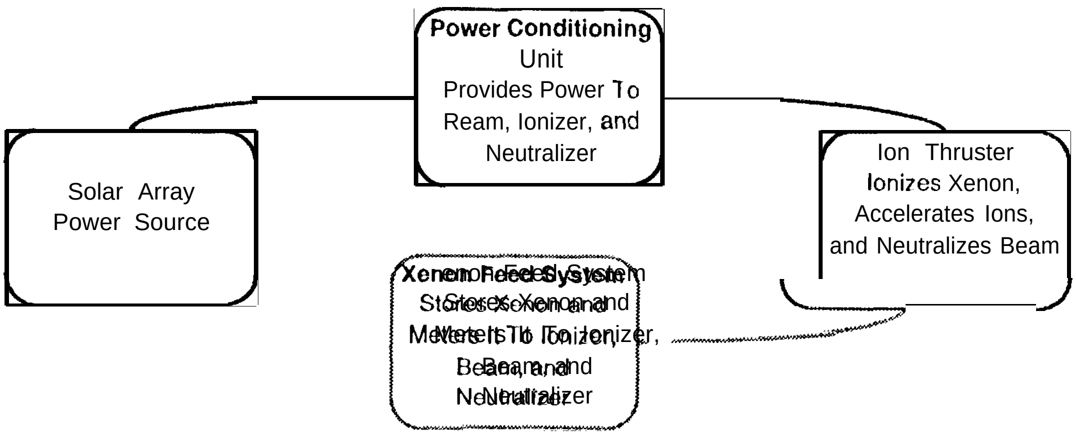


Figure 13. Conceptual block diagram of ion propulsion system.

4.1.3 *Objectives*

The NSTAR program will accomplish the following objectives:

- Ensure that ion propulsion technology meets pertinent mission requirements by basing validation requirements on missions of interest.
- Validate life, integration, and performance in a ground-test program.
- Measure in-space interactions with the spacecraft and the surrounding space plasma by flying an ion propulsion experiment on a host spacecraft.
- Stimulate commercial sources for and uses of solar-powered ion propulsion.

4.2 NASA's **Empirical Approach**

4.2.1 *Validation Approach*

To provide the information a Project Manager needs to baseline ion propulsion on a spacecraft, it is first necessary to determine what information is required. After this information has been identified, it is then necessary to demonstrate empirically that the hardware can satisfy the requirements.

The process for determining NSTAR requirements was accomplished in two stages. In the first stage, which continues at a low level, the user communities were surveyed to identify each community's needs. User needs were then ranked and taken as requirements (Ref. 7).

User-based requirements were then carefully apportioned to various tests and experiments that make up the NSTAR validation program. The major requirements are shown in Table 3, in which each user requirement corresponds to an NSTAR test or experiment addressing that requirement. The tests that comprise the ground-test element foeus on the key issues that must be considered in any application of electric propulsion. The ground test will determine the following:

- Demonstrate service life including the modeling necessary so that the data taken during life testing can be applied to a spectrum of missions,
- Demonstrate performance (power handling, thermodynamic efficiency, specific impulse), and

- Demonstrate integrability, i.e., the ground measurement of EMI and plume effects.

The in-space experiments address key issues that can only be determined in space. The in-space experiments will accomplish the following:

- Measure direct effects (e.g., contaminant ion, EMI) on the spacecraft and surrounding space plasma,
- Measure indirect effects that influence the cost of electric propulsion missions (e.g., guidance, navigation, and control (GN&C) and MO&DA), and
- Determine whether data taken during ground tests accurately replicate the data obtained during in-space operation.

The NSTAR program is executed jointly by the Lewis Research Center (LeRC) and the Jet Propulsion Laboratory, taking advantage of the best experience, facilities, and expertise available within NASA. LeRC is **responsible** for providing the ion thrusters and power processing units for the groun[l-test element and for the in-space experiment. JPL manages the program rrnd is responsible for developing program requirements, the xenon feed system, and the in.space diagnostics,

The validation tests shown in '1'able 3 include ground-based tests that will determine ion engine life and performance and will measure plume transmissibility and EMI. The in-space experiment will measure the effects of ion propulsion on a Spacecraft as well as cm the host spacecraft and the surrounding space plasma. The in-space experiment will also assess the ability of ground testing to replicate the data obtained during operation in space. In-space operation will further demonstrate the capability to integrate and operate an ion propulsion system.

4.2.1.1 Schedule

Figure 14 shows a schedule for ttrc two parallel elements: a ground-test element and an in-space experiment element.

4.2.1.2 Ground-Test Element

In the ground-test element, the first parallel element, lifetime and performance of the system will be demonstrated, and data necessary for integration of the ion propulsion system will be collected for plume divergence, plume transmissibility, and EMI.

The grouncl-lest element is composed of four main tests and several supporting test series. Three engineering model thrusters and two breadboard power processors will be used in the test program.

The first engineering model thruster will be used in a 2,000-hour test to confirm whether the life-limitin~ mechanisms, principally erosion of the accelerator grid by charge-cxchan.gc ions, arc the same as those observed in past versions of the 30-cm ion thruster. Furthermore, this test should provide the most accurate measurement to date of the wear-rrut rates associated with the **various wear-mrt mechanisms**, Upon completion of the 2,000-hour test, the thruster will be refurbished and then subjected to a series of environmental qualification tests; these tests will serve as precur-

Table 3. Summary of NSTAR validation requirements.

Requirement	NSTAR Test Addressing Requirement	Planned Test/Experiment Date
Thruster lifetime of 8,000 hr with demonstrated margin of 50%	Life Validation Test	1996 1997
Thruster cyclic life equivalent to 15-years station keeping, and 2 repositions pcr year	Cyclic Life Test	1996
Assessment of wear-orrt mechanisms and determination of their rates	Cyclic Life Test Life Validation Test	1996 1996 1997
Power processor efficiency of 92% at maximum power	Cyclic Life Test Life Validation Test	1996 1996 1997
Demonstration of ion propulsion system integration with host spacecraft	System Integration	1998
Commercial source for ion propulsion flight experiment	Delivery and integration of Flight Thruster anti Power Processor	1999
Measurements of in-space performance of ion propulsion system and comparison to ground test results	In-Space Experiment	1999 2000
Measurements of ion propulsion system interactions with host spacecraft and surrounding space plasma (contamination, EMI, communications, etc.)	in-Space Experiment	19992000
Assessment of impacts of ion propulsion on GN&C and MO&DA	1'0s1 Irl-Space Experiment	2000

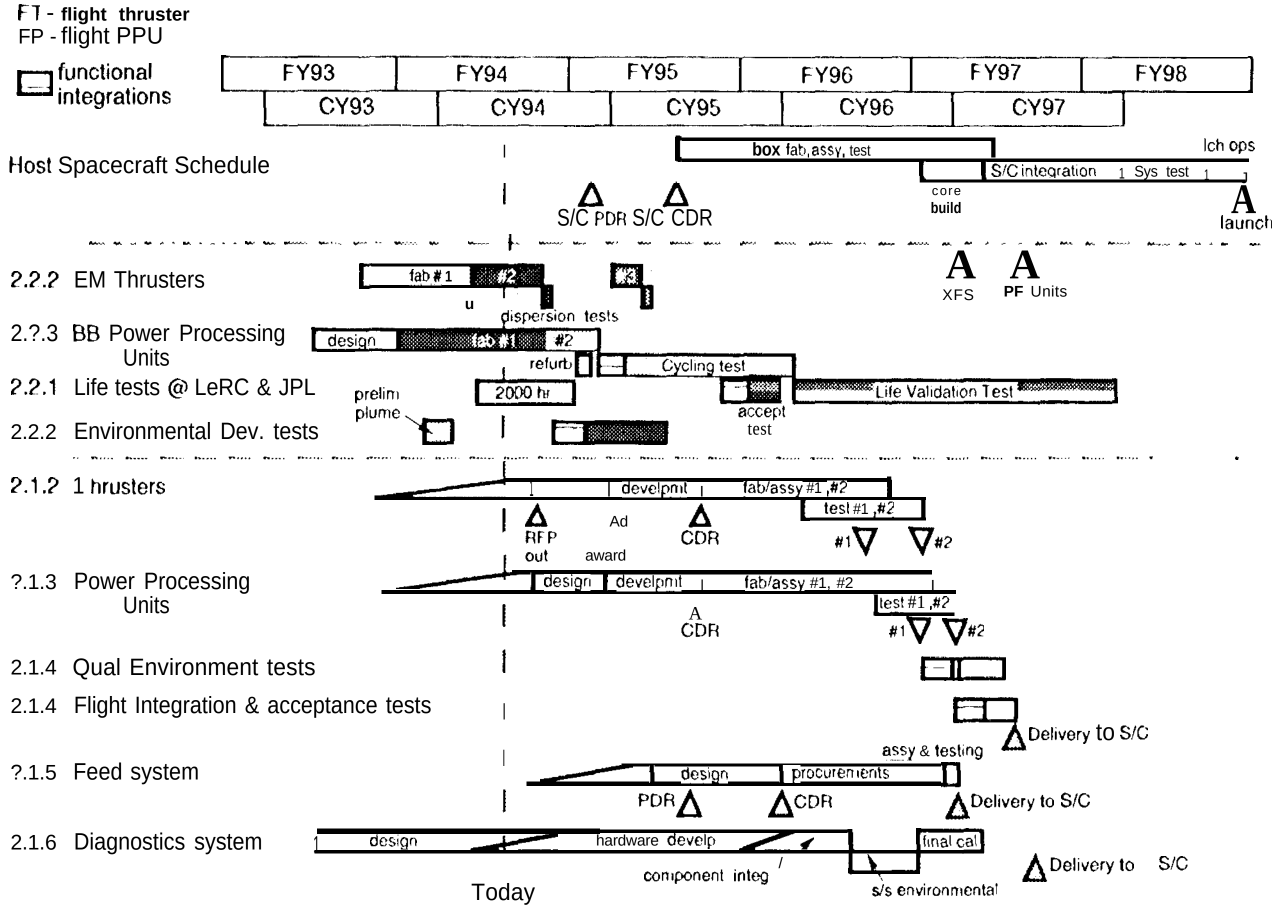


Figure 14. NS1 AR activity schedule.

sors to a similar series of tests that will be conducted cm a protoflight thruster.

The second engineering model thruster and the first breadboard power processor will be subjected to a test designed to simulate a thruster performing station keeping and repositioning for a satellite in geosynchronous orbit. This test will be conducted at full power for a duration of 5,700 hours and a total of S,000 on/off cycles. At specific points during the test series, approximately once each month, the thruster's performance over its entire throttling range will be tested.

The third engineering model thruster and the second breadboard power processor will bc used in the Life Validation Test, a long-duration test to validate that the service life of the system is more than 8,000 hours. This test requires a minimum of 12,000 hours that will demonstrate a 50% margin relative to the desired 8,000-hour service life. The full 2.5-kW input power will be used for the duration of the test, except when periodic measurements of performance at throttled conditions are performed, which will be approximately once per week. The validation of thruster life will bc considered successful after 12,000 hours have been demonstrated. The test will continue until the thruster ceases to perform acceptably, thereby providing additional data for life assessment and failure analysis.

One of the principal objectives of these tests, besides verifying performance and demonstrating acceptable life, is identifying all significant life-limiting processes and quantitatively measuring the pertinent rates associated with each process.

4.2.1.3 In-Space Element

The second parallel element is the in-space experiment, in which the interaction of the ion propulsion system with the host spacecraft and the surrounding space plasma wilt be investigated and quantified. The ion propulsion experiment was designed as part of the USAF/TRW ELITE spacecraft. An artist's conception of the in-space configuration of the experiment is shown in Figure 15.

The in-space element is intended to accomplish several objectives. The first objective is to demonstrate in-space performance and capabilities of ion propulsion. The measurements taken during the course of the in-space experiment will quantify the direct effects of ion propulsion on the host spacecraft and the surrounding space plasma, Indirect impacts of ion propulsion will also be assessed. Indirect impacts include changes to spacecraft integration activities — as compared to chemical propulsion — changes in the conduct and execution of guidance, navigation, and spacecraft control, changes in the planning, training, and conduct of MO&DA, and changes in the scheduling of other spacecraft activities.

The direct measurements taken in orbit will include

. Contamination, particularly of optical and cooled surfaces,

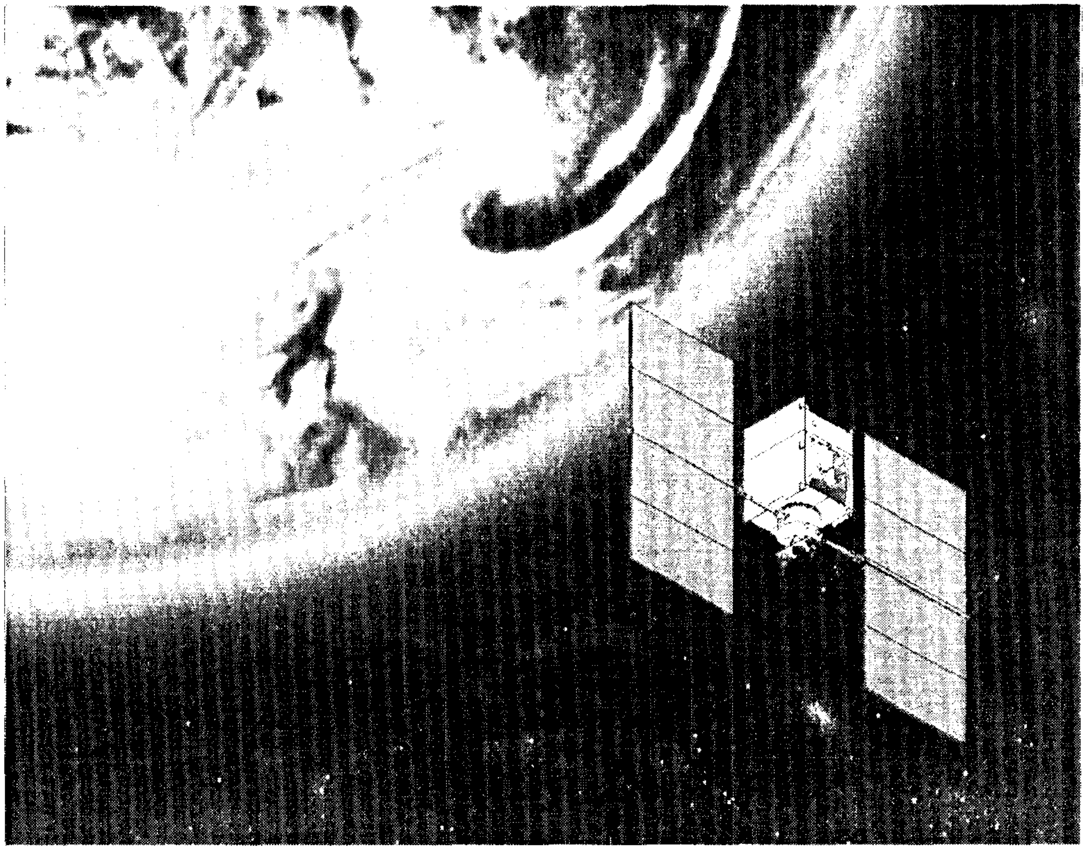


Figure 15. ELITE spacecraft with NST AR ion propulsion experiment.

- Communications, particularly of transmission through the plume of tbc ion engine,
- EMI, particularly steady-state and transient-induced magnetic and electric fields, and
- Effects of the ion propulsion system on the electric and magnetic properties of the surrounding space plasma, Particular attention will be paid to obtaining data necessary to assess the effect of an ion propulsion system on interplanetary field and particles measurements.

The physical measurements that will accomplish the direct measurements arc described in "1'able 4.

Measurements taken during operation of the ion thruster in ground testing will be compared with data obtained during in-space operation. For example, an integration of feed system pressure data combined with careful tracking of the spacecraft will enable researchers to estimate thruster performance. Power and thermal measurements will allow power conversion efficiency to be determined. Such gross measurements arc important to confirm tbc adequacy of measurements taken during ground testing.

Also important arc measurements of parameters that arc supposedly influenced by the hard vacuum of space, such as accelerator grid impingement current. The behavior of these parameters as a function of thruster-crpcrating condition and duration will be studied carefully.

After tbc mission, indirect impacts will be assessed by examining the change in the costs and degree of difficulty caused by incorporating ion propulsion on spacecraft integration and system test, GN&C, MO&DA, and scheduling.

The it]-space element will address the program's objective to stimulate commercial sources for and uses of ion propulsion. For all government users, this objective is important for several reasons. 1 f no commercial source of ion propulsion technology is avail able, then ion propulsion technology cannot bc incorporated on government spacecraft. If a commercial source exists but no commercial uses of the technology arc made, then the costs of this technology to the government would be significantly greater than would be the case were commercial users available.

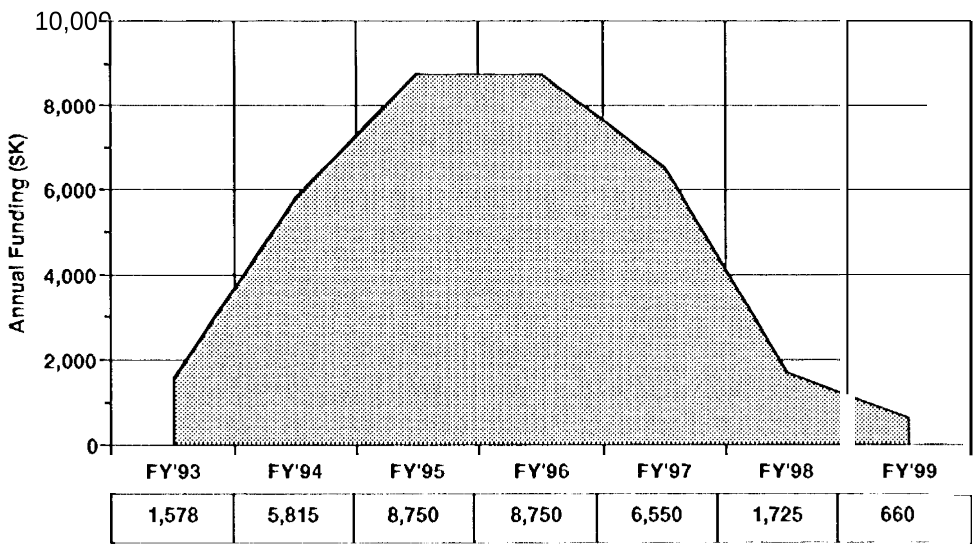
The two flight ion thrusters and the two flight power processors will be purchased from a commercial source. Wc expect that the commercial source will participate in NASA's ground-test element. This participation should provide NASA's industrial partner with the knowledge and hands-on experience needed to continue the technical evolution of the ion propulsion system after NSTAR is completed. NASA would then turn its attention to the next generation of propulsion equipment, just as it did following the successful infusion of the hydrazine arcjet into the commercial space sector in 1993.

4.2.1,4 Funding

The funding profile planned for the NSTAR validation program is shown in Figure 16. These funds arc equally split between the ground-test portion of tbc program and the in-space portion and

Table 4. In-space diagnostics measurements for the NSTAR space experiment.

Instrument	EMI	Plasma or Spacecraft	Plume	Communication	Contamination	Radiatio
Electric Field Antenna	✓					
Langmuir Probe		✓	✓			
Spacecraft Potential Probe		✓				
Internal Discharge Monitor	✓					
Solar Array Current Collectors		✓	✓			
Magnetometer	✓					
Mass Spectrometer					✓	
SGLS Omni Antenna				✓		
X-band Transmitter/Receiver				✓		
Quartz Crystal Microbalance					✓	
Calorimeter					✓	
Optical Effects Monitor					✓	
Solar Photovoltaics					✓	✓
Radiation Monitor						✓
Microelectronics						✓



Total Funding = $33.8 M

Figure 16. Funding profile.

include a nominal allowance for the integration of the NSTAR in-space experiment onto the host spacecraft. These expenditures *do not* include the cost of civil service personnel from LeRC, who support NSTAR, nor do they include the cost of the solar array, which it is assumed will be provided with the host spacecraft.

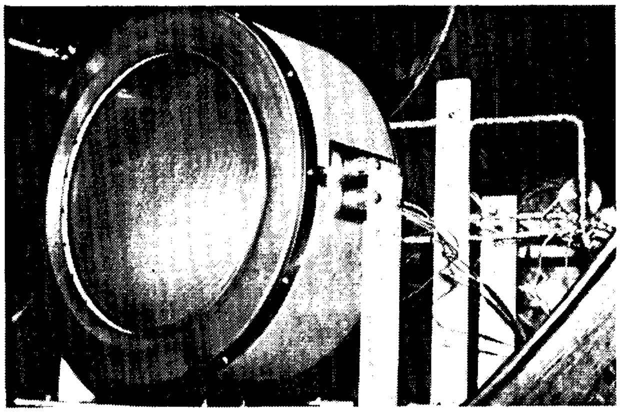
4.3 Status of NSTAR Technology Validation Program

4.3.1 Ground-Test Program

The ground-test element of the NSTAR validation program has been under way since late 1993, when testing of the functional model thruster was begun. The purpose of this test was to provide data that would verify a design in which the mass of the ion thruster is reduced to 7 kg, making it more producible. These tests were successfully completed early in 1994 and served as a precursor to the subsequent fabrication and testing of the first engineering model thruster.

Figure 17 shows this 30-cm thruster installed in the test facility prior to the test. Figure 18 shows the neutralizer installed on the thruster prior to the test. The preparation of the LeRC 15-foot-diameter-by-60-foot vacuum chamber is shown in Figure 19. At the bottom of the figure, the inlets for the diffusion pumps can be seen. Figure 20 shows a hollow cathode before it was installed on the first engineering model thruster. The hi@-voltage isolator used on the main cathode feed]inc is shown in Figure 21, and the feed system flow controllers installed outside the vacuum chamber are shown in Figure 22.

On June 23, 1994, the 2,00(-hour test for the first engineering model thruster began and is continuing at the time of this meeting. During testing, a malfunction in the facility power supply resulted in a hiatus of several weeks; this occurred after 870 hours of the test had been completed without incident. The purpose of the test was to confirm earlier work that identified wear-out mechanisms and to quantify the rates associated with those mechanisms. Preliminary examination of the data from the 2,000-hour test confirms that the principal wear-orrt mechanism is the erosion of the accelerator gr id by charge-cxchangc ions, indicating that the expected life of the thruster is comfortably in excess of the 12,000 hours required to demonstrate an 8,000-hour service life.



F igure 17. Engineering model thruster-1 in place for 2,000-hour wear test.

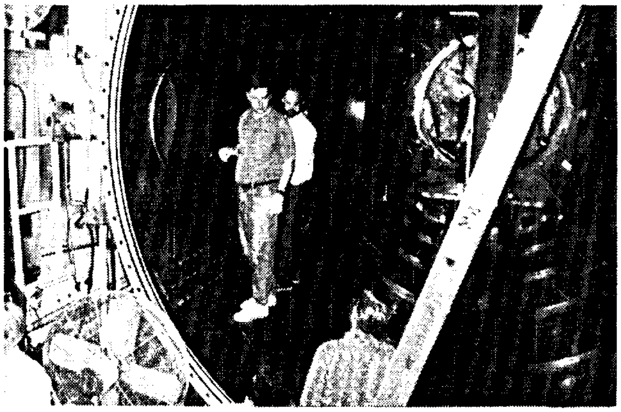


Figure 19. Vacuum test facility.

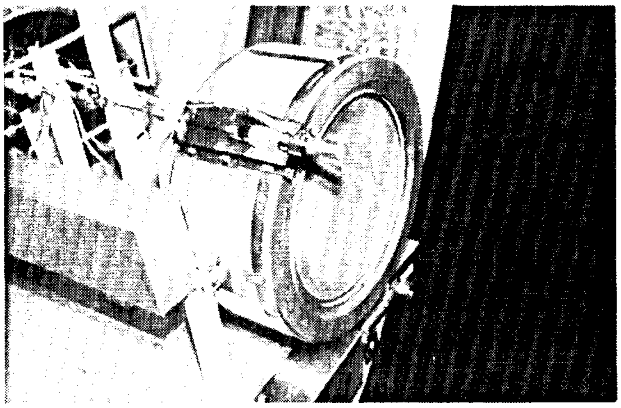


Figure 18. Engineering model thruster-2 showing neutralizer.

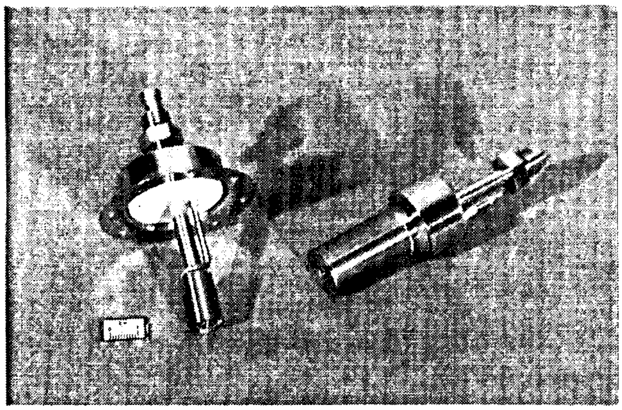


Figure 20. Hollow cathodes used for main discharge (left) and neutralizer.

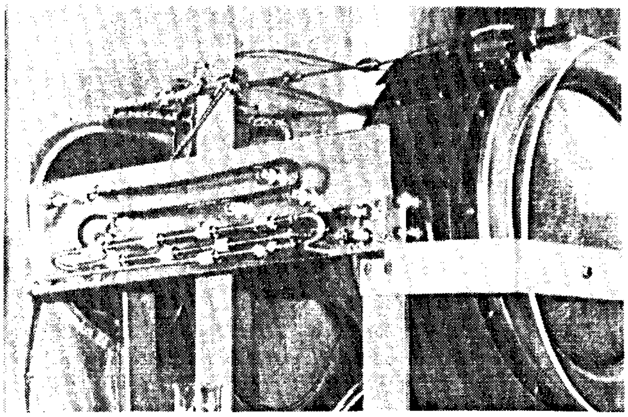


Figure 21. High-voltage isolator with metal box cover (shown in Figure 18) removed.

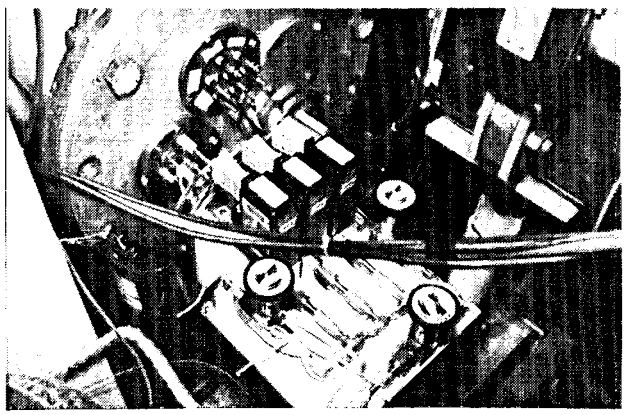


Figure 2?. Facility xenon flow-control system mounted adjacent to and outside of the vacuum chamber.

The second engineering model thruster is being fabricated, and the first breadboard power processor is nearing completion anti should be ready for the Thruster Cycling Test planned for April 1995. The first engineering model thruster will be refurbished and used for simulated environmental qualification tests, verifying the ability of the future flight thruster to withstand the rigors of the protoflight qualification program.

4.3.? *Host Spacecraft*

Early in 1994, the cooperative partnership between TRW, inc. and the USAF/Phillips Laboratory that was to result in the ELITE program was effectively ended due to cuts in U.S. defense spending. With it, the initial impetus for the validation of ion propulsion technology using a host spacecraft for the in-space expel-imcnt clement also ended. However, NASA believed the benefits of validating low-power ion propulsion technology for future government and commercial missions were significant and decided to continue the NSTAR validation program and to redouble efforts to find a host spacecraft that would support the ion propulsion experiment. As of this writing several opportunities have been identified, both with the USAF and NASA. Present planning has focused on the first or second integrated Space Technology Flight (ISTF) planned by USAF/Phillips Laboratory.

5 CONCLUSIONS

- Typically, the number of arcjet spacecraft maneuvers are 1 to 2 times that of chemical propulsion, and ion engine maneuvers are 2 to 10 times the number provided by chemical propulsion for the range of spacecraft masses and powers considered here.

- Power requirements for the ion propulsion arc greater than that for the arcjet, but not much greater for spacecraft maneuvers up to 1 0 deg/day. Therefore, ion propulsion is the best choice when the mission priority is to increase the number of spacecraft maneuvers at moderate maneuver rates.

- ESEX, a program conducted by the USAF, provides the first on-orbit validation and demonstration of the technology associated with high-power arcjets.

- NASA has initiated a program to validate low-power ion propulsion technology called NSTAR.

- The NSTAR validation program consists of two mutually dependent, interlocking parts: a series of ground-based tests and an in-space experiment.

- Successful completion of the NSTAR validation program will significantly improve the performance and life of large satellites in GEO, for both civilian and military use, by reducing the mass required for on-orbit station keeping and repositioning.

- Successful completion of the NSTAR validation program will significantly improve the performance of NASA's small comet rendezvous and planetary flyby missions.

- Successful completion of the NSTAR validation program will result in the development of a commercial source for ion propulsion flight equipment.

6 ACKNOWLEDGMENT

The authors wish to acknowledge and thank the staff of the USAF/Phillips Laboratory's Propulsion Directorate for their contribution and assistance in the completion of this paper.

The research described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration and at Lewis Research Center and the USAF/Phillips Laboratory.

7 REFERENCES

1. Sutton, A. M., "Overview of the Air Force ESEX Flight Experiment," IEPC-93-057, 23rd IEPC, September 1993.

2. Aadland, R. S., et al., "Integrated Mission Simulation of a 26 kW Arcjet Flight Arcjet Propulsion System," AIAA 93-2395, June 1993.

3. Deininger, W. D., "30-kW Ammonia Ar~.jet Technology," Final Report, July 1986 December 1989.

4. Johnson, L. K., et al., "Frequency-Domain Electromagnetic Characteristics of a 26 kW Ammonia Arcjet," AIAA 93-2393, June 1993.

5. Kriebel, M. M., Stevens, N. J., "30-kW Class Arcjet Advanced Technology Transition Demonstration (ATTD) Flight Experiment Diagnostic Package," AIAA 92-3561, July 1992.

6. Turner, J. B., "The Advanced Research and Global Observation Satellite (ARGOS) Program," AIAA 94-4580, Space Programs and Technologies Conference, September 1994.

7. Kakuda, R. 1'., et al., *Small Body Rendezvous Mission Using Solar Electric Ion Propulsion: Low Cost Mission Approach and Technology Requirements*, IAA International Conference on Low-Cost Planetary Missions, IAA-L-0710, April 12-15, 1994.